Mars Global Surveyor Mission

S. Sam Dallas
Jet Propulsion Laboratory
California Institute of Technology
Pasadena, CA 91109
818-393-1263
Sateri ios.S.Dallas@jpl.nasa.gov

Abstract - The Mars Global Surveyor (MGS) Spacecraft will be launched in November of 1996 I() the planet Mars for an extended study of the planet's surface, atmosphere, gravitational field and magnetic field. 11'1 order to achieve the scientific objectives of" the mission, the spacecraft will be inserted into a low-altitude, near-polar, sun-synchronous orbit. Data will be collected and returned from six prim experiments on the spacecraft for over one Martian year (687 Earth days) and will provide for a better understanding of the geology, geophysics, and climatology of" Mars.

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1. INTRODUCTION

NASA and the Jet Propulsion Laboratory will begin America's return to Mars after a 20-year absence by launching the Mars Global Surveyor (MGS) spacecraft in November of 1996, The MG S mission will collect data on the surface, atmosphere, gravitational field, and magnetic field of the red planet over a two-year period. The MGS spacecraft with its scientific instruments will be p laced in a low-altitude, near - polar, Still-syllclllol)olls" orbit around Mars. This paper will describe the mission design, the science payload, the spacecraft, and the launch vehicle for this mission.

?. SCIENCE PAYLOAD

During 687 days of mapping operations at Mars, the science payload will generate more than 600 orbits of raw data to achieve the five basic goals of the mission:

(1) Characterize the surface morphology at high spatial resolution,
(2) Determine the global elemental, thermophysical, and mineralogical character of the surface material,
(3) Define the global topographical and gravitational field,
(4) Establish the nature of the magnetic field, and
(5) Monitor the global weather and thermal structure of the

atmosphere 10 evaluate their seasonal impact on the polar caps, atmospheric dust and clouds.

In addition, the data will support mission and scientific planning for future Mars expeditions with special emphasis on the selection of possible future landing sites. The names of the scientific instruments for MGS, the location of the principle investigator and his home institution, and the measurement objectives for each instrument are given in'J'able 1.

Table 1. Science Instruments Summary

Acronym	Full Name	Principle Investigator and Home Institution	Objective
Mag/ER	Magnetometer and Electron Reflectometer	M.H. Acuna Goddard Space Flight Center (GSFC)	Intrinsic magnetic field and solar wind interactions with Mars
MOC	Mars Orbiter Camera	M.C. Malin Malin Space Systems (MSSS)	Surface and atmospheric imaging
MOLA	Mars Orbiter Laser Altimeter	D.E. Smith Goddard Space Flight Center (GSFC)	Surface topography and gravity field studies
MR	Mars Relay Radio System	J. Blamont Centre Nationale d'Estudies Spatiales (CNES, France)	Support for future Mars missions, both American and International
TES	Thermal Emission Spectrometer	P. R. Christensen Arizona State University (ASU)	Mineralogy, condensates, dust, thermal properties, and atmospheric measurements
USO (RS)	Ultra Stable Oscillator for Radio Science	G.L. Tyler Stanford University (team leader)	Gravity field determination and atmospheric refractivity profiles

Magnetometer and Electron Reflectometer (MAG/ER)

Mars is the only planet from Mercury to Neptune whose magnetic field has not yet been measured. The design of the solar-array-mounted magnetometers will allow for the in-situ measurement of the global magnetic field in three perpendicular direction over an extremely wide dynamic range between 16 and 65,536nT. Given this data, the electronic reflectometer will determine the local lwar-surface magnetic fields by measuring the deflection of ambient electronics (energy range of 1 -20 keV) by the local magnetic field. Because the paths of these electrons are significantly altered by the magnetic field, use of the electron reflectometer in conjunction with the magnetometer will provide for a resolution capability 10 to 100 times greater than using the magnetometer alone.

Mars Orbiter Camera (MOC)

This instrument consists of two independent cameras mounted onto a Single assembly. Both cameras are supported by a 32-bit microprocessor for data acquisition and compression, and a 12 Mbyte buffer for temporary image storage. The narrow-angle camera employs a 70-cm tall, f/10 Ritchey - Critien reflector with a focal length of' 3.5 meters. Inside, two 2048-element charged couple device (CCD) line arrays sit (m the focal plane and are mounted it) a direction perpendicular to the spacecraft's velocity vector during mapping operations. Two dimensional images of the Martian surface at an unprecedented resolution of 1.4 m/pixel will be formed as the motion of the spacecraft sweeps the detectors forward ,

The (X)101'- capable, wide-angle camera consists of two f6, 9.7 mm focal length, 140° field of view fish-eye lenses feeding into a single focal plane containing two 3456 -element CCD line arrays. In order to produce color images, the two lenses use red (5-/5 - 625 nm) and blue (400 - 500" nm) filters, respectively. Wide-aup,le images of the surface with a resolution of 250 m/pixel at nadir (? km/pixel at the limb) will be produced in the same "motion swept forward'" fashion as the narrow-angle images. '1'hese wide-angle pictures will contribute to the MOC's global monitoring mode, an experiment that will provide daily, fall-planet observations of the Martian atmosphere and surface similar to the weather pictures of Earth shown during newscasts.

Mars Orbiter Laser Altimeter (MOLA)

The Mars Orbiter Laser Altimeter (MOLA) experiment will generate high-resolution topographic profiles of Mars for studies of geological structures and processes. This goal will be accomplished by using a diode-pumped neodymium-yttrium aluminum-garnet laser that will fire 4 S-rid pulses of light at the Martian surface at a rate of 10 bursts per second. By recording the time that the pulse takes to reach the Surface and bounce back to the instrument's 50-cm Cassegrain collecting mirror, the MOLA team will be able to compute the local altitude under the spacecraft along the ground track.

Each laser spot will measure about 160 meters in diameter (m the surface with a spacing of about 300 meters between spots along the ground track. The accuracy in measuring relative topography will vary from one to 10 meters, with an absolute accuracy of about 30 meters. Ultimately, the absolute accuracy will depend on the precise post-reconstruction of the spacecraft orbit position from navigation and radio-science data.

Mars Relay (MR)

The Mars Relay (MR) consists of a radio system and antenna designed to return measurements and imaging data from spacecraft deployed on the sill'face of the red planet. 'Ibis instrument consists of a 1 -meter tall helix antenna mounted on the nadir panel of the Spat'm"af[and all of the associated elect ronics. Unlike the main X-band communications system , this device operates at UI IF frequencies. The antenna pattern (-3 db) takes the form Of a 65° cone emanating from the tip of the antenna, providing coverage with a 5,000 km effective range for a 8 kbps data rate, and a 1,300" Jim range for a 1 28 kbps rate. As the spacecraft orbits Mars, the M R will transmit a 1.3-W, 437.1-Mhz beacon to the surface, indicating, to the landers that the MGS spacecraft is currently in view. '1 'his beacon will serve as an indicator for the landers 10 begin transmitting their data.

Radio Science (RS)

Radio-science experiments will advance two field fundamental to the study of Mars. First, observations of distortions (frequency, phase, and amplitude) in the spacecraft's rad io signal as it passes through the Martian atmosphere 011 the way to Earth will be used to derive high-resolution temperature profiles of the atmosphere with a vertical resolution of 200 meters. Second, by using Doppler tracking 10 carefully monitor small changes in the. frequency of the radio signal from the spacecraft as it orbits Mars, the radio- science team will be able to reconstruct the Martian gravity field to an unprecedented level of accuracy, perhaps higher than a S() x 50 field.

Both radio-science experiments will require precise tracking of the spacecraft's radio signal by the antennas of NASA's Deep Space Network. 111 order to facilitate this requirement, the spacecraft will employ an ultra- stable oscillator (USO) to provide an extremely stable frequency reference for the X-band telecommunications system. This high-quality, low-noise oscillator resonates at 19.143519 Mhz and will have a long-term frequency variation limit of less than 1 .0 x 10]0 Hz.

Thermal Emission Spectrometer (TES)

The Thermal Emission Spectrometer (TES) will function as a combined infrared spectrometer and radiometer designed to measure heat energy radiated from the surface and atmosphere of Mars. The investigation team will use data from '1'1 ES 10 determine the thermal and mineralogical proper ties of the sill'raw, and to learn about Martian atmospheric properties, including cloud type and dust opacity.

This instrument primarily consists of two nadir-pointing telescopes. The larger of the two is a 15.24-cIII diameter Cassegrain design that feeds a two-port Michelson interferometer spectrometer with a spectral range from 6.?5 10 so pin. The smaller of the two serves two bolometric channels (0.3 to 3.9 μm and 0.3 to I 00 μm) and takes the form of an off-axis parabola shaped telescope. Each telescope utilizes six detectors, each with an 8.3 x 8.3 mrad field of view. Together, the six form a rectangular grid three frames wide (cross-track) and two frames deep (down-track). Although the instrument will normally remain nadir pointed, a rotatable scan mirror will allow the TES telescopes to view Mars at any arbitrary oblique angle from horizon to horizon.

3. SPACECRAFT

Lockheed Martin Astronautics built the Mars Global Surveyor spacecraft at their Denver facility. When fully loaded with propellant at the time of launch, the Mars Global Surveyor spacecraft will weigh no more than 1,062 kilograms under the current design and ΔV budget. In order to meet this target mass, the spacecraft structure consists of lightweight composite material divided into four sub-assemblies known as the equipment module, the propulsion module, the solar arrays, and the communication antennas (See Figure 1).

Equipment Module

The equipment module houses the avionics packages and science instruments. The dimensions of this rectangular shaped module measure 1.221 x 1.221 x 0.762 meters in the X, Y, and Z directions, respectively. With the exception of the Magnetometer, all of the science instruments will be bolted to the nadir equipment deck, mounted above the equipment module on the +Z panel. The Mars Relay antenna is the tallest instrument and extends up 1.115 meters about the nadir equipment deck.

Two redundant flight computers inside the equipment module will orchestrate almost all of the spacecraft's flight activities. Although only one of the two units will control the spacecraft at any one time, identical software will run concurrently in the backup unit. Each computer control unit consists of a Marconi 1750A microprocessor, 128 k RAM for storing command sequences as

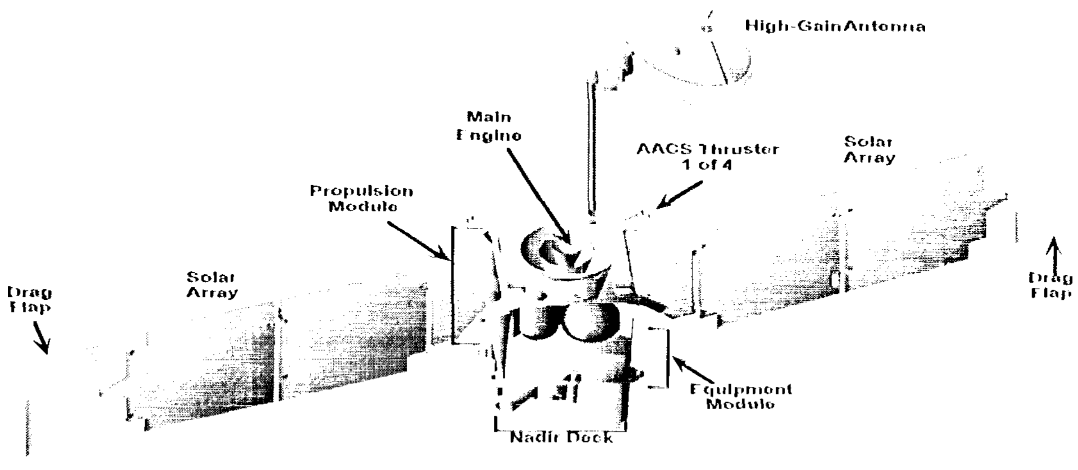

Figure 1. MGS Spacecraft

long as six weeks in duration uploaded from Earth, and 20 K PROM that contains code to run basic survival sequences upon entry into fault protection mode due to anomalous conditions." Additional storage space for science and engineering data will be provided by two solid-state recorders, each with a 1,500 Mbit capacity. The MGS mission will represent America's first interplanetary spacecraft to exclusively use RAM instead of a tape recorder for mass data storage. This technology improvement will dramatically reduce operational complexity thereby reducing mission planning costs during flight.

The equipment module also contains three reaction wheels mounted in orthogonal directions to provide spacecraft pointing control authority for all mission events, except for major propulsive maneuvers. A fourth wheel mounted in a direction skewed to the other three will serve as a redundant unit. Attitude information for the spacecraft will be provided by Sun sensors, an inertial measurement unit, Mars horizon sensors, and a star scanner.

The spacecraft's 25-watt RF power amplifiers inside the equipment module will provide the capability for downlink of science and engineering telemetry at data rates between 21,333 Sps to 85,333 sps, depending on the varying Earth to Mars distance. The "sps" stands for symbols per second, and a symbol is essentially a Reed-Solomon encoded (250:218 ratio) bit. Therefore, it takes approximately 1.147 bits of storage space to encode one bit of raw data with this encoding ratio. Spacecraft communications with Earth will always utilize X-Band frequencies for radiometric tracking, return of science and engineering telemetry, commanding, and radio science experiments. However, the spacecraft's telecommunications equipment also accommodates Ka-band carrier-only downlink for the purposes of providing a feasibility demonstration.

Propulsion Module

The propulsion module serves as the adapter between the launch vehicle and the spacecraft and contains the nitrogen tetroxide (NTO) and hydrazine tanks, main engine, propulsion feed system, and attitude control thrusters. This module bolts beneath the equipment module on the -Z panel and consists of a rectangular shaped box 1.063 meters on each side, with a 0.310 meter tall cylinder shaped launch vehicle adapter extending from the bottom of the box. Each corner of the box portion of the module contains a small metal protrusion that houses attitude control thrusters. Including the length of these protrusions, the diagonal widths of the propulsion model measure 2.464 and 2.394 meters long.

The main engine, used for large maneuvers such as major trans-Mars trajectory corrections (TCMs) and Mars orbit insertion (MOI), will burn a bi-propellant combination of NTO and hydrazine, and will deliver an I_{sp} of 315 to 318 seconds at a thrust level of 659 N. During main engine burns, four rocket-engine modules (REMs), each containing three 4.45 N thrusters (two aft facing, and one for roll control), will burn hydrazine in mono-propellant, pulse-on mode to provide attitude control. In addition, these mono-propellant thrusters will also be used in pulse-off mode for small trajectory corrections during cruise and orbit trim maneuvers at Mars, and for unloading momentum from the reaction wheels. This propulsion system differs from a conventional bi-propellant system in that the same hydrazine tank will serve both the main engine and attitude-control thrusters, rather than using a separate hydrazine tank for each system. The propulsion system will provide the MGS spacecraft with a ΔV capability of 1,282 m/s. This budget assumes a spacecraft dry mass of 6"/3 kg and a total mass of 1,062 kg.

Solar Arrays

Two solar arrays, each 3.531 meters long by 1.854 meters wide, will provide energy for the MGS spacecraft. Each array mounts close to the top of the propulsion module on the -1 Y and - Y panels, near the interface between the propulsion and equipment modules. Including the adapter that holds the array to the propulsion models, the tips of the arrays extend 4.270 meters from the sides of the spacecraft. Rectangular shaped, metal drag "flaps" mounted on the ends of both arrays add another 0.813 meters to the overall array structure. These "flaps" serve no purpose other than to increase the total surface area of the array structure to increase the spacecraft's ballistic coefficient during aerobraking.

Each array consists of two panels, an inner and outer panel comprised of gallium arsenide and silicon cells, respectively. Available power will start at 1,100 W immediately after launch. During mapping operations at Mars, this amount will vary from a high of roughly 940 W at Mars perihelion to about 660 W at aphelion. When the spacecraft moves into eclipse or turns away from the Sun, energy will flow from two nickel-hydrogen (NiH$_2$) batteries, each with a capability of ?0 Amp-hours.

Communication Antennas

Primary communications to and from the spacecraft will occur through the 1.5-meter diameter high-gain antenna (HGA). From launch until the start of mapping operations at Mars, the HGA will remain body fixed to the spacecraft on the +X side of the spacecraft. Consequently, using the HGA will require slewing the spacecraft to point directly at the Earth. During mapping operations, the HGA will be deployed and will sit at the end of a 2.0 meter boom mounted to the +X panel of the propulsion module. This configuration will allow the HGA to automatically track the Earth by means of two single-axis gimbals that hold the antenna to the 1300111.

In addition to the HGA, the spacecraft also carriers four low-gain antennas (LGA) for emergency communications. Two of the LGAs will function as transmit antennas, while the other two will receive. Placement of these four LGAs will ensure that the spacecraft can receive commands and downlink telemetry over a wide range of attitude orientations. The primary transmit LGA is mounted on the HGA, while the backup is mounted on the +X side of the propulsion module. The two receive LGAs are mounted on the -X pm] of the equipment module and the +X side of the propulsion module.

4. LAUNCH VEHICLE

Launch of the Mat's Global Surveyor spacecraft will utilize the McDonnel Douglas Delta 11 7925A launch vehicle (see Figure 2). The Delta II 7925A launch vehicle consists of five major components that include the three main stages, a set of nine solid rocket motors that attach to the first stage, and the payload fairing. The Delta measures 38.2 meters tall from the tip of the payload fairing to the bottom of the first stage's body. Including the spacecraft the Delta will weigh approximately 231, 327 kg at the time of launch.

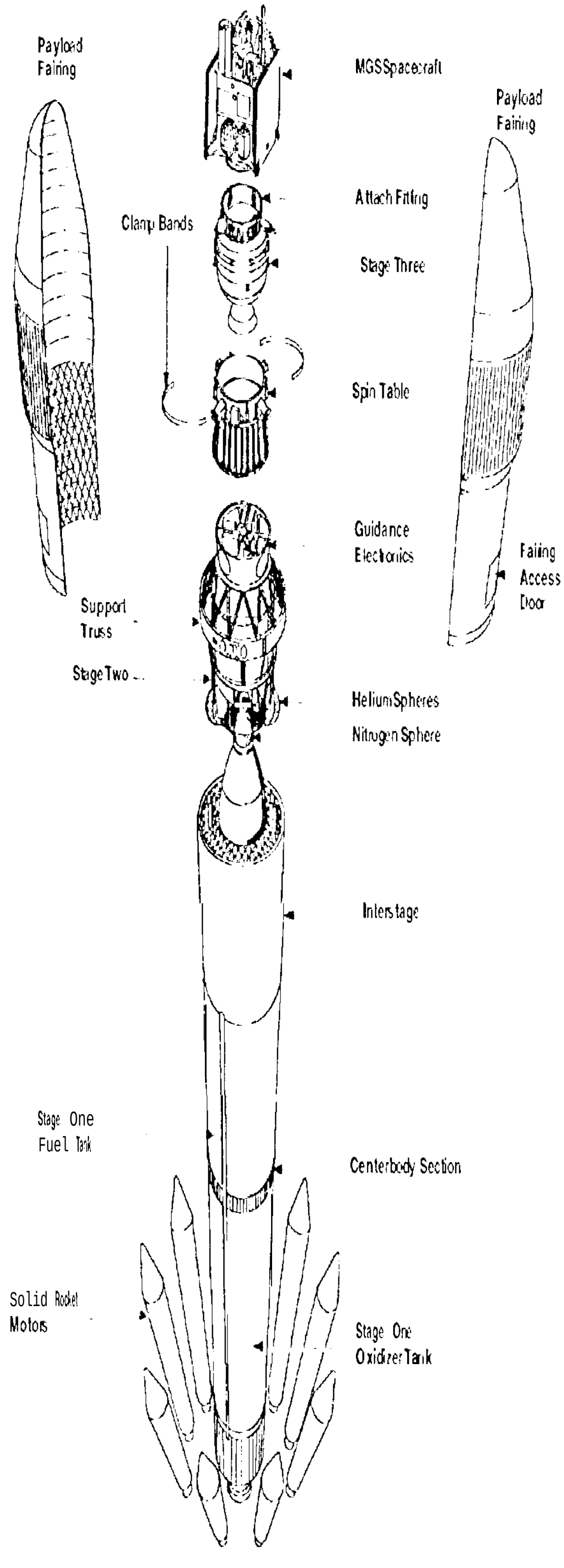

Figure 2. Delta II 7925A Launch Vehicle

Stage one employs a Rocketdyne RS-27A main engine with a 12:] expansion ratio. This engine is a single start, liquid, bi-] ›ropel lant rocket that will provide nearly 890,000 N of thrust at the time of lift-off. 11s propellant load (95,655 kg) consists of 1<1'-1 fuel (thermal ly Stable kerosene) and liquid oxygen (I .OX) for oxidizer. The 1<1'-1 fuel tank and liquid oxygen tank on the first stage are separated by a center body section that houses control electronics, ordnance sequencing, equipment, a telemetry sy stem, and a rate gyro. In addition, stage one also employs Iwo Rocketdyne vernier engines. 'J 'hey wil i provide roll control during the main engine burn, and attitude control between main engine cutoff (MI :CO) and second stage ignition.

A set of nine solid-propellant graphite epoxy motors ((i] :Ms), each fueled with approximately 1 2,000 kg of hydroxyl - terminated polybutadiene (11'1'1']) solid propellant, attach to the first stage to provide augmentation thrust. 1:ach (i] :M will provide an average thrust of 446,000" N. Six of the nine GI:Ms, the main engine, and the vernier engines will ignite at the time of lift- off, producing a total thrust of about 2,850,()()0 N. The remaining three (i] :Ms will ignite 65 seconds into flight, shortly after the initial six burnout.

The 1)elta second stage uses a re-startable, liquid, bi-propellant Aeroject AJ 10-118K engine that consumes a combination of Aerozine-50 fuel (a 50/50 mix of hydrazine and un-symmetric dimethyl hydrazine) and nitrogen tetroxide (N_2O_4) oxidizer. Since this pı opellant combination is hypergolic, no catalyst oı ignitei in the engine thrust chamber is required. in total, the second stage will burn nearly 6,000 kg of propellant at an average thrust of 43,3'70 N. A set of hydraulically activated engine gimbals will provde pitch and yaw control during powered flight, and a nitrogen cold gas jet system will provide the roll authority. in addition, the nitrogen jets will also provide attitude Control for the coast phases.

A spin- stablized third stage consists primaı ily of a Thiokol Star-48B solid motor. The engine on the third stage will provide an average thrust of 66,370 N and will burn about 2,000" kilograms of ammonium perchlorate propellant.

During launch and ascent through the lower atmosphere, a 2.9 meter diameter payload fairing will protect the spacecraft and Delta third stage from aeı odynamic heating. The fairing will be jettisoned from the launch vehicle at an altitude of approximately 129 kilometers (L+ 286 seconds), shortly after second stage ignition.

5. MISSION PHASES

Five mission phases have been defined in [1] to describe the different periods of activity. These are the launch, cruise, orbit insertion, mapping, and relay phases. Table 2 summarizes the dates of the mission phases and some key mission events. The dates in 'Table' 2 are specific to the mission that launches at

he opening of the launch period on 6 November 1996. Under the current launch period strategy, MGS can launch as late as 25 November, 1996. Figure 3 shows the general mission timeline, the Martian seasons and science data rates.

Launch Phase

After lift-off, the first stage of the three-stage Delta rocket will boost the spacecraft to an altitude of 115 km. From there, the second stage will take over and achieve a 185-km, circular, parking orbit at about 1-1 10 minutes, After parking orbit insertion, the booster and spacecraft will coast for between 24 and 37 minutes (variable with launch date) until reaching a position over the eastern Indian or western Pacific ocean. At that time, stage two will re-start and thrust for nearly two minutes to mist the apogee of the parking orbit. Then, spin

Table 2. Mission Phases and Maneuvers

Event	Date	Comments
Launch	6 Nov 1996 (early afternoon EST)	Launch period opens on 6 Nov and closes on 25 Nov 1996
Inner Cruise Phase	6 Nov 1996 to 6 Jan 1997	Communications through LGA only because solar arrays must be pointed at a fixed angle to the sun
TCM1	20 Nov 1996 (L+14 days)	Trajectory correction for injection errors, remove aim-point biasing introduced for Mars planetary quarantine
Outer Cruise Phase	7 Jan 1997 to 11 Sep 1997	Communications through HGA, phase begins when Earth-MGS-Sun angle falls below 60°
TCM2	21 Mar 1997 (TCM1+120 days)	Correct for execution errors from TCM1
TCM3	20 Apr 1997 (TCM2+30 days)	Correct for execution errors from TCM2
TCM4	22 Aug 1997 (MOI-20 days)	Final adjustment to MOI aim point
Mars Orbit Insertion (MOI)	11 Sep 1997 (about 01:15 UTC)	MOI can vary from 11 Sep 1997 to 22 Sep 1997 depending on exact launch date
Orbit Insertion Phase	11 Sep 1997 to 14 Mar 1998	Begins at MOI, lasts 5 months to reach mapping orbit using aerobraking and propulsive maneuvers
Mapping Phase	15 Mar 1998 to 31 Jan 2000	Mars mapping operations for one Martian year, about 687 Earth days in duration
Relay Phase	1 Feb 2000 to 1 Feb 2003	About 3 years of support for future Mars missions

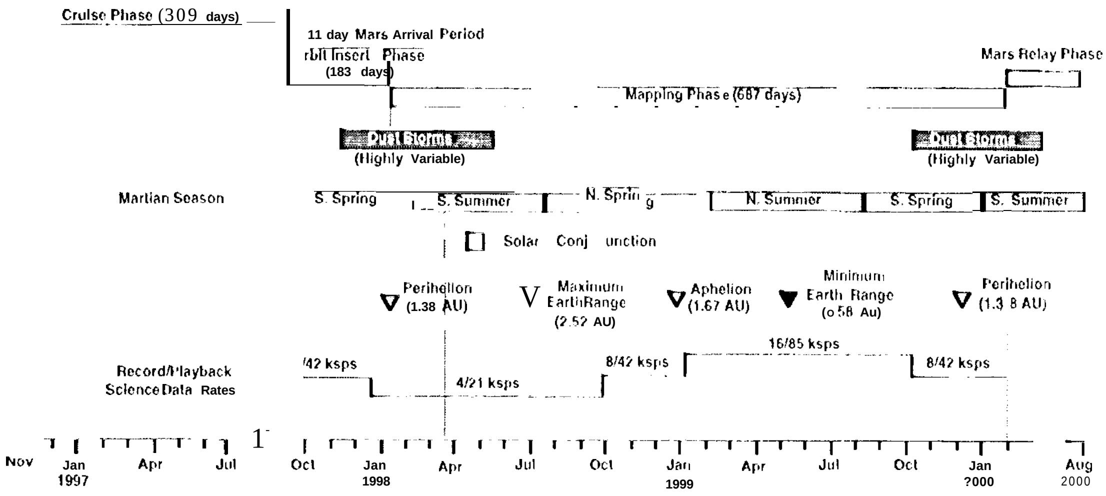

Figure 3. Mission Timeline

rockets will spin-up the third stage and spacecraft to 60 r.p.m., followed by third stage ignition. The Delta's third stage, a STAR 48B solid, will fire for 87 seconds to complete the trans-Mars injection (TMI) burn. After completing the TMI burn, but before third stage jettison, a yo-yo cable device will deploy from the STAR 48B to de-spin the spacecraft. Figure 4 shows the MGS launch flight profile.

Cruise Phase

Cruise covers the time of ballistic flight between Earth and Mars. The spacecraft will take between 301 and 309 days to reach the red planet on its Type-2 trajectory depending on the Earth departure date within the 20-day launch period. A launch at the open of the launch period on 6 November 1996 will **correspond** to a Mars arrival date of 11 September 1997, while a launch at the close of the period on 25 November 1996 will result in an arrival on 22 September 1997. During cruise, a set of four trajectory correction maneuvers (TCMs) will adjust the interplanetary trajectory to ensure that the spacecraft reachs the proper velocity and position targets prior to the Mars orbit insertion (MOI) burn. Figure 5 shows the MGS cruise flight profile.

During the first part of cruise, called inner cruise, initial deployment and checkout of the spacecraft will be accomplished, and navigation tracking data will be taken to determine the flight path for the purpose of planning and executing the first of four planned trajectory correction maneuvers (TCMs). TCM1 is scheduled to occur 14 days afer launch (-L14 days).

In inner cruise, all spacecraft communications with the Earth will occur through the low-gain antenna (LGA). The reason is primarily due to the spacecraft configuration and solar panel geometry. Because the high gain antenna (HGA) sits on the spacecraft in a stowed, body-fixed orientation during cruise, communicating with the Earth through HGA will require turning the spacecraft to point the antenna directly at Earth. However, such an orientation would push the incidence angle of sunlight on the panels past acceptable levels for minimum power generation. Therefore, communications through the LGA represents the only feasible option.

Outer cruise will begin when the spacecraft witches from use Of the low gain to the high gain antenna for communications with the Earth. The exact time when the switch becomes feasible depends on whew the angle between the Sun and earth as seem from the spacecraft (SPE) falls to a level low enough to allow good power while the spacecraft is oriented to point the HGA directly at Earth. This angle starts at about 1200 at the time Of launch and falls to less than 60° by 6 January 1997, assuming a launch on 6 November 1996.

Currently, the transition date to switch to the HGA from the LGA will occur on 6 January 1997 for the purpose of planning command sequences. However, this date will be subject to change during flight as the spacecraft team evaluates the telemetry. In the interest Of maintaining the highest possible communications link margin with Earth, switchover will occur as early as possible.

Most of outer cruise will consist of minimal activity as the spacecraft transits to Mars. The vast majority of the events will involve acquiring navigation and tracking data to support the remaining TCMs. During the last 30 days Of approach to Mars, the focus will be On final targeting of the spacecraft to the proper aim point, and preparations for orbit insertion. A series of Mars approach images will be taken during this "mars approach" time period using the Mars orbiter carom. Figure 6 presents

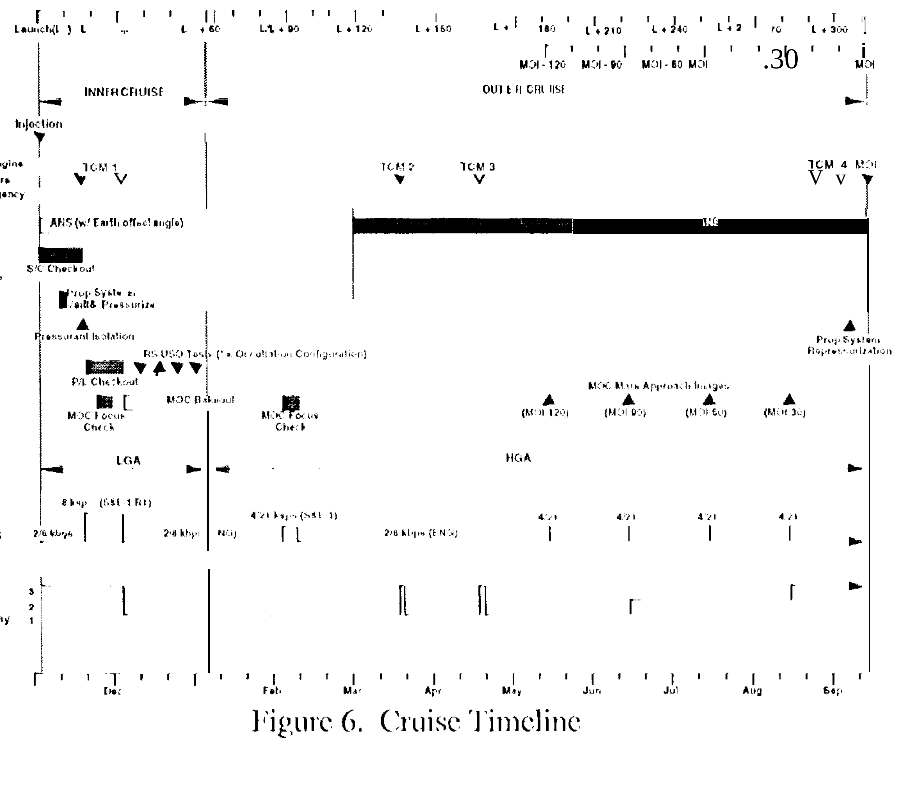

Figure 6. Cruise Timeline

the MGS cruise timeline.

Orbit Insertion Phase

During approach, the spacecraft's velocity relative to Mars will be approximately five kilometers per second. Near the periapsis of the inbound hyperbolic trajectory, the 596-N main engine will fire for between 20 to 25 minutes to provide a ΔV of about 980 m/s. Burn ignition will occur about 10 minutes before periapsis. During the burn, the spacecraft will utilize a "pitch-over" maneuver to slew the spacecraft at a constant rate in order to keep the thrust nearly tangent to the trajectory arc. After burn cut-off 10 minutes after periapsis, the spacecraft will orbit Mars in a highly elliptical orbit with a period of 48 hours and a periapsis altitude of about 300 km (periapsis radius of 3,700 km). Figure 7 shows the Mars approach trajectory profile.

The MGS spacecraft will not carry enough propellant to propulsively reach the required low-altitude, Sun-synchronous mapping orbit due to the relatively low interplanetary injected mass capability of the low-cost Delta launch vehicle. Consequently, the spacecraft will rely on aerobraking, an innovative mission-enabling technique, to trim the initial, highly-elliptical, capture orbit down to mapping orbit altitudes. During aerobraking, the spacecraft will pass through the upper fringes of the Martian atmosphere on every periapsis pass. Friction from the atmosphere during the drag pass will cause the spacecraft to lose a small amount of energy and will cause the altitude on the next apoapsis pass to slightly decrease. The rate at which the apoapsis altitude decreases will be determined by the amount of drag generated. Aerobraking deeper in the atmosphere will provide greater drag and reduce the orbit faster, but will generate higher spacecraft temperatures and dynamic pressures.

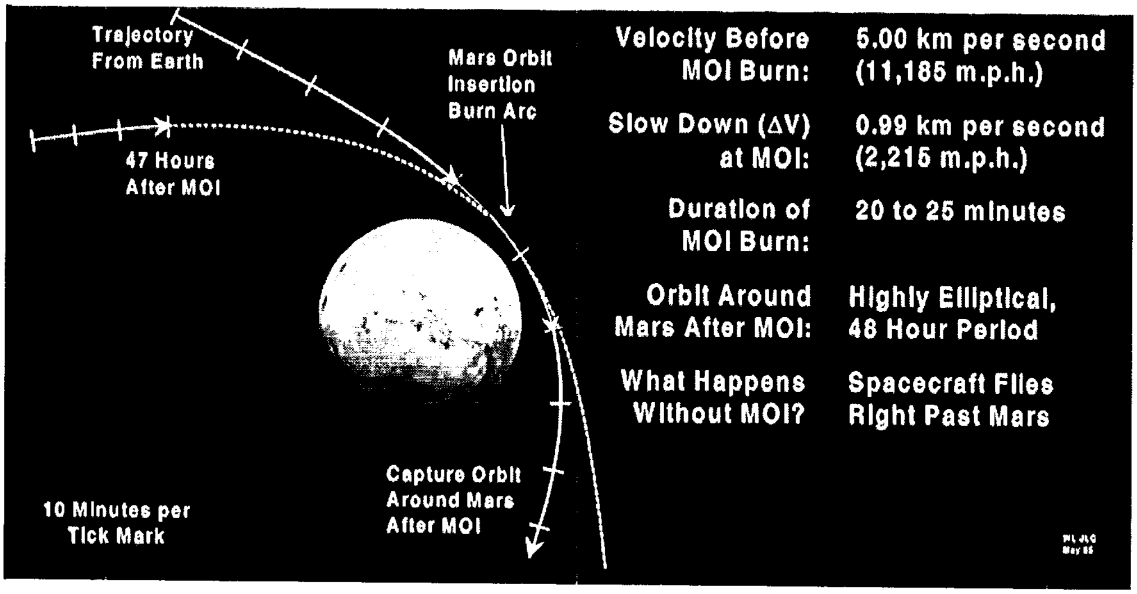

Figure 7. Mars Approach Trajectory Profile

Aerobraking will begin nine days after the oribt insertion burn with the first of four maneuvers (AB1) designed to lower the periapsis altitude into the Martian atmosphere in four gradual steps. The AB1 burn will be the largest of the four and will lower the periapsis to 150 km. The next three (AB2, AB3, and AIM) will provide a further drop to 112 km. This need for a gradual walk-in is due to the large uncertainty in the atmospheric density model of Mars, and will allow the navigation team time to study the atmospheric density and its orbit-to-orbit variation. All four walk-in burns will be performed using the attitude control thrusters.

After completion of walk-in, the spacecraft will spend about three months in the main phase of aerobraking. During this phase, the apoapsis altitude will shrink in size from about 56,000 km down to 2,000 km. As needed, small propulsive maneuvers (ABMs) executed at apoapsis will maintain periapsis within a well-defind periapsis altitude corridor low enough to produce enough drag to reduce the orbit within the time constraints to reach the proper orbit orientation relative to the sLm, yet high enough to avoid spacecraft heating and maximum dynamic pressure limits. Due to the oblateness of Mars and the fact that periapsis will be migrating northward toward the pole during main phase, the altitude Of periapsis will tend to rise. Consequently, most of the ABMs will be in the down direction to lower the periapsis altitude into the control corridor.

The three weeks of aerobraking following the main phase will represent an extremely critical period as the altitude Of apoapsis is lowered toward the final altitude Of 450 km. During this time, the spacecraft will be slowly "walked-out" Of the atmosphere by gradually raising its periapsis altitude to 143 km. Daily ABMs will be performed as necessary to maintain a guaranteed wmt-case, three-day mbit lifetime. Hence, in the absence of ABMs due to unforeseen events that inhibit the ability of flight controllers to send commands, the spacecraft will always be at least three days from crashing into the surface.

Aerobraking will end with a termination burn (ABX) performed sometime during mid-January 1998. This burn will raise the orbit periapsis Out Of the atmosphere to an altitude Of approximately 450 km. At this time, the spacecraft will be circling in a 400 x 450 km orbit with a period slightly under two hours. In addition, the descending node location will have regressed from its original MOI position at 5:45 p.m. with respect to the fictitious mean Sun to the. desired 2:00 p.m. position. Figure 8 shows the various phases of aerobraking.

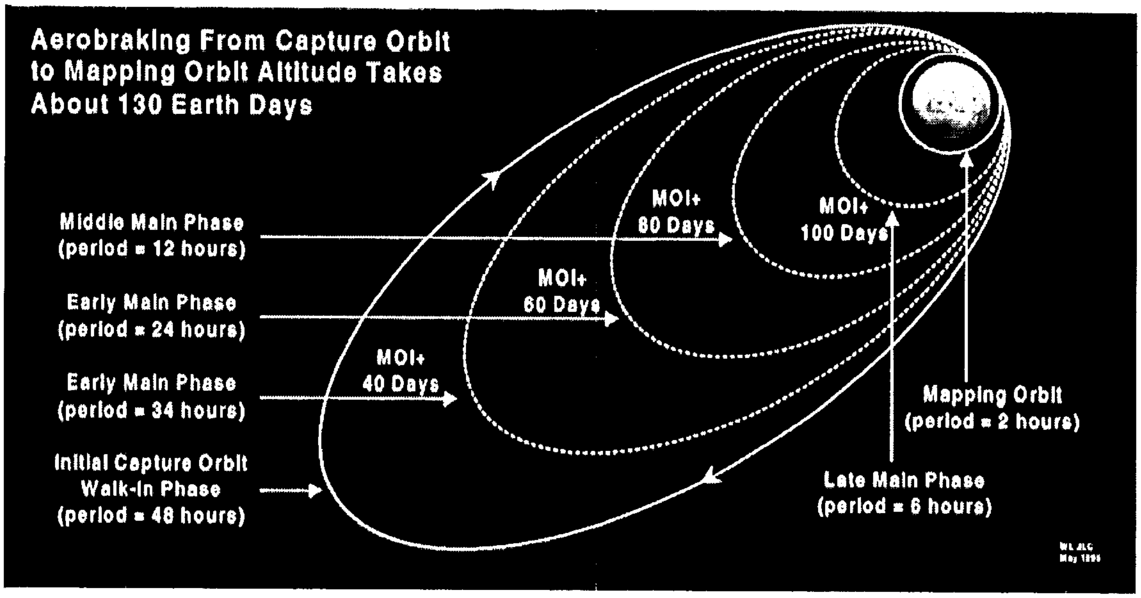

Figure 8. Phases of Aerobraking

Transition-to-mapping will begin at the end of aerobraking and will last until the final mapping orbit has been established and the spacecraft is declared ready to begin mapping operations. After the aerobrake termination burn (ABX), the spacecraft will circle Mats in a transition mbit for a month. During this waiting period, the oblateness of Mars will alter the mbit and cause the location of periapsis to drift to a position almost immediately above the Martian South Pole. At that time, currently scheduled for mid-February 1998, the transition-to-mapping orbit (TMO) burn will be performed with the intent Of "freezing" the periapsis location at the South Pole, and establishing the proper altitude for mapping operations. Throughout this entire transition period between ABX and TMO, the navigation team will conduct gravity calibrations to update the Martian gravity field model using in-flight navigation data returned from the red planet. This update will be crucial toward accurately executing the TMO and other future burns.

An mbit-trim maneuver (OTM) burn will be executed to refine the frozen orbit 12 days after TMO. The 12 days is driven by the navigation team's need for four days to track the spacecraft after the TMO burn and another eight days to plan the maneuver. After the OTM, a ten-day spacecraft deployment and checkout period will follow to allow the operations team time to configure the spacecraft and its instruments for mapping operations. Figure 9 shows the mbit insertion timeline.

Mapping Phase

The mapping phase represents the period of concentrated return of science data from the mapping orbit. This phase will start on 15 March 1998 and last until 31 January 2000, a time period Of one Martin year (687 Earth days). These dates will remain fixed and are independent of the actual day Of lift-off

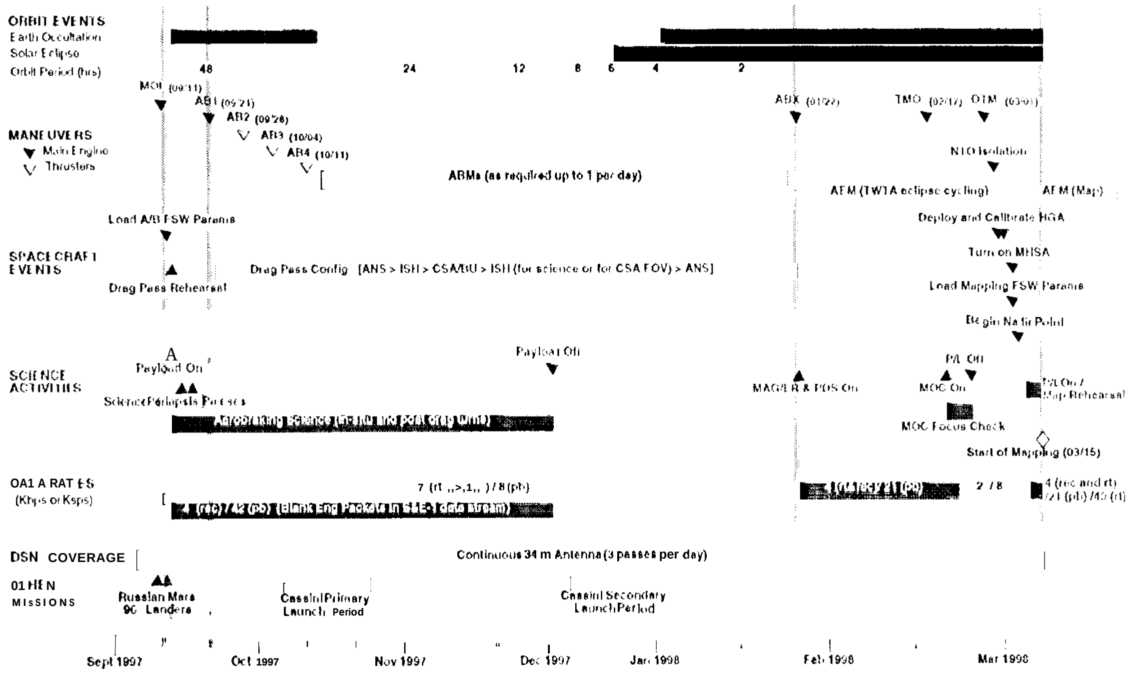

Figure 9. MGS Orbit Insertion Timeline (Open Launch Period)

within the launch period. During this phase, the spacecraft will keep its science instruments (-Z panel of the spacecraft) nadir pointed to enable data recording on a continuous basis. On a daily basis, the spacecraft will transmit 24 hours of recorded data back to the Earth during a single 10-hour Deep Space Network (DSN) tracking pass. An articulating high gain antenna (HGA) on the spacecraft will allow data recording to proceed while downlink to Earth is in progress. Figure 10 shows the MGS mapping orbit.

Relay Phase

The relay operations phase will begin at the end of mapping and continue for *three* years. During this phase, the spacecraft will function as a relay satellite for various Mars landers in support of the Mars Exploration Program. End of mission will occur on 1 February 2003.

6. PROJECT STATUS

The MGS spacecraft is currently on the Delta II 7925A launch vehicle at Space Launch Complex 17A, Cape Canaveral Air Station in Florida ready for launch. The MGS mission operations teams located at JPL and LMA are in their final prepartions for launch.

REFERENCES

[1] Wayne Lee, Mars Global Surveyor Project Mission Plan Document, MGS 542-405, Final Version, Rev. A, Jet Propulsion Laboratory, July, 1996.

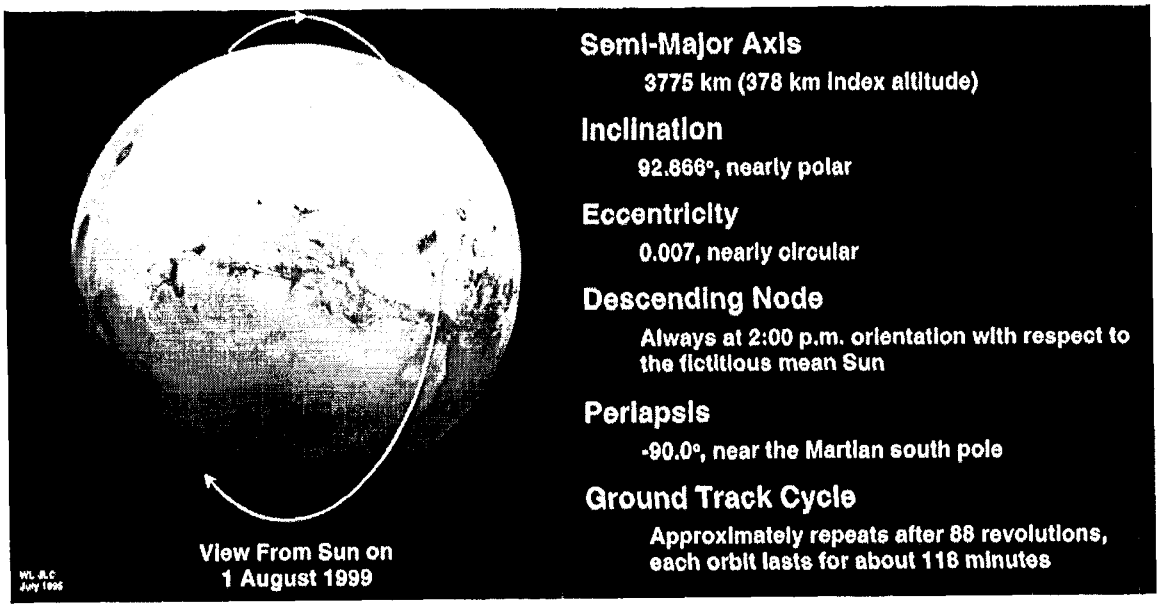

Figure 10. Mapping orbit

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Wayne Lee	Mission Plan (JPL)
John Callas	Mission Plan (JPL)
Dan Johnston	Trajectory Design (JPL)
Dan Lyons	Aerobraking Design (JPL)
Wayne Sidney	Mission Timelines (LMA)

Dr. Saterios Sam Dallas received a B.S. in Aeronautical Engineering (1959), and a B.S. in Engineering Mathematics (1960), from the University of Michigan. He then received a M.S. in Engineering (1963), and a Ph.D. in Engineering (1968), with a major in Astrodynamics from the University of California.

Dr. Dallas was employed as an engineer for JPL in June of 1959. Since that time, he has been involved in research associated with the motion of artificial and natural celestial bodies. He has written many articles and reports on this subject.

Currently, Dr. Dallas is the proposal manager for a long range science rover mission to Mars. The completed proposal will be submitted to the NASA Discovery Program.

Prior to this assignment, Dr. Dallas was the mission manager for the Mars

Global Surveyor Project. in this capacity, he had the responsibility of managing the mission design and mission operations development activities for the Project.

Prior to this assignment, Dr. Dallas was the Mission Manager for the Mars Observer Project. In this capacity, he had the responsibility of managing the mission design and mission operations activities including the Science Operations Teams, the Spacecraft Team, the Navigation Team and the Sequence Team for the Project.

Prior to this assignment, Dr. Dallas was the Science and Mission Design Manager for the highly successful Magellan Project. In this capacity, he had the responsibility of managing the science planning activities, the Navigation Development Team, and the Mission Design Team for the Project.